Low Acceleration Rate Ignition for Spacecraft

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A g-dot ignition system for solid-propellant motors has been designed to prevent damage to fragile sensors or structural members on a spacecraft by producing, from a starting transient for the spacecraft of 0.3 g, a controlled buildup in thrust, such as to give an acceleration rate of about 0.3 g/s. The system consists of a 3-s, regressive-burning, controlled-flow igniter and a highly inhibited progressive-burning charge in the main motor. The igniter must operate in a hard vacuum and sustain burning of the uninhibited portion of the propellant below its normal L^* combustion limit through mass addition and heat transfer until the propellant surface has increased sufficiently to provide a stable motor chamber pressure. A specific internal torus-shaped igniter is described along with its advantages and disadvantages as well as potential methods of initiating burning in the hard vacuum with Pyrofuze.

Introduction

Current work on low-thrust, long-burning solid-propellant motor technology is oriented primarily toward orbit insertion missions at Jupiter, Saturn, and perhaps Mercury, because of extensive scientific interest and a recent JPL study (unpublished) of outer-planet orbiter missions. Figure 1 shows an artist's rendering of a Jupiter orbiter spacecraft from that study.

Note the long, highly flexible appendages for the scientific instruments and for the radioisotope thermoelectric generator on the opposite side of the spacecraft. These dictate a maximum spacecraft acceleration tentatively set at 1 g until trade-off studies can be made, and imply, inherently, low thrusts and long-burning times for outer-planet orbit insertion motors. Of special concern to the effort under discussion, the flexible appendages also dictate low acceleration rates, or g-dots, during the starting and shut-down thrust transients if limit cycling of the guidance components is to be avoided.

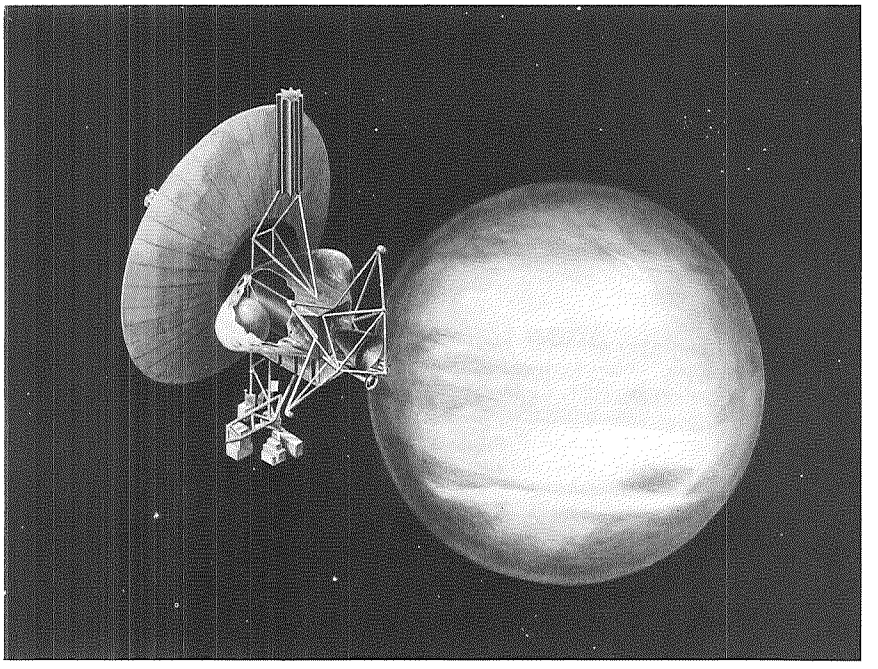

Figure 1. Jupiter orbiter spacecraft

A low spacecraft acceleration rate during thrust initiation will be achieved through the use of an unconventional *g*-dot ignition system that will gradually build up thrust at a controlled rate of about 0.2 to 0.3 *g*/s from its starting value, about one-third to one-fifth the maximum acceleration level. Reference 1 discusses some *g*-dot ignition feasibility studies and tests.

The low spacecraft deceleration rate of 0.2 to 0.3 *g*/s during thrust decay would be provided through a propellant charge geometry that automatically produces a gradual thrust tailoff, i.e., through propellant sliver formation.

Operation of the *g*-Dot Ignition System

Figure 2 shows an artist's rendering of the 363-kg (800-lb)[1] demonstration motor, designated D-2, in which the torus-shaped *g*-dot igniter will be used.

Operation of the igniter system can best be explained by reference to Figure 3, where the motor D-2 is shown in cross section with the torus-shaped *g*-dot igniter mounted on the submerged portion of the nozzle. The igniter is actually a small solid-propellant motor which burns with

[1]Where applicable, the International System of Units is stated first, followed by the customary units in parentheses. In each case, the value in parentheses represents the measured unit.

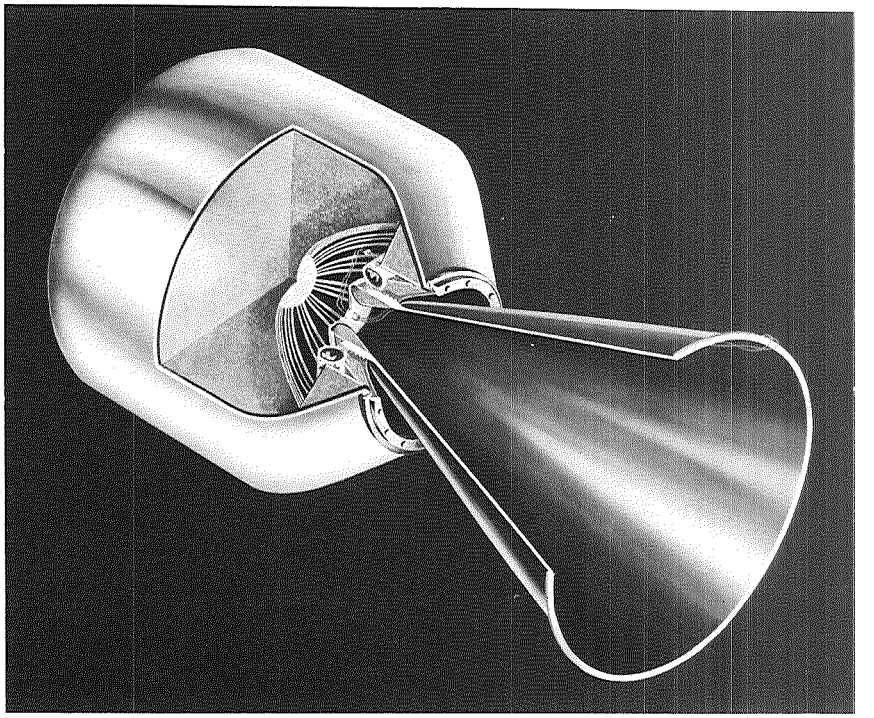

Figure 2. Demonstration motor D-2

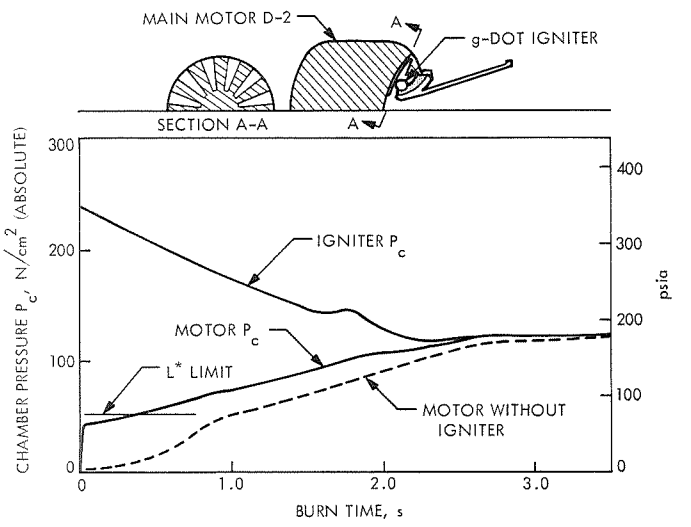

Figure 3. The *g*-dot ignition concept

supersonic exhaust for about 1½ s of its 2½-s burning time. Its combustion gases pass radially outward from numerous nozzles, impinging on the dish-shaped propellant surface. Igniter pressure (absolute) decreases from about 241 N/cm² (350 psi) to about 124 N/cm² (180 psi).

The main motor propellant surface is highly inhibited, as shown in section A-A, Figure 3, with an inhibiter pattern such as to produce a highly progressive burning surface as the propellant regresses under the inhibitor surface. Thus, if the motor propellant could burn at a very low pressure by itself, its pressure–time curve would resemble the broken line in Figure 3—with an initial pressure (absolute) of only 3.45 to 6.89 N/cm² (5 to 10 psi).

In reality, because of the L^* combustion limit of about 44.8 N/cm² (65 psi) absolute, the motor would not burn by itself below that pressure. However, when hot exhaust gases from the independent controlled-flow igniter are injected into the main motor, the mass addition raises the motor pressure (absolute) to about 34.5 to 37.9 N/cm² (50 to 55 psi) and burning of the main charge is sustained below the motor L^* limit by the heat transfer and mass addition. The resultant low pressure and thrust level permit the spacecraft to meet its 0.3-g initial acceleration requirement.

The main propellant burning surface and, consequently, chamber pressure increase with time in a controlled manner until the motor is able to sustain combustion without mass addition from the igniter. The small, thin inhibiter strips are partially, or completely, consumed before being ejected out of the nozzle.

The pressure–time relationships shown in Figure 3 represent nominal performance values. In practice, however, operation of the main motor below its normal L^* combustion limit has produced low-frequency low-amplitude oscillations characteristic of L^* combustion instability near the extinction pressure (Reference 1).

Concern about potentially unacceptable acceleration rates from these oscillations prompted a recent series of computer runs in which the effect of the oscillations on the spacecraft attitude controls was simulated using the Thermoelectric Outer-Planet Spacecraft model, modified to include the retro-propulsion for a Jupiter orbiter mission. It was concluded that, for both rigid and flexible body dynamics, the oscillations arising from the use of the *g*-dot igniter and other low-amplitude bumps and spikes which may occur in the thrust profile had no apparent effect on the attitude control system.

In the same series of computer runs, the effect of initial acceleration during igniter initiation and the acceleration rate during igniter burning on the spacecraft controls was also evaluated. The following results were obtained:

Maximum initial acceleration, *g*	Acceleration rate during igniter burning, *g*/s	Effect on spacecraft controls
0.2	0.2	Acceptable
0.4	0.4	Acceptable
0.6	0.6	Limit cycling

Because tests were not all-encompassing in their scope, conservative values of 0.3-*g* maximum acceleration and 0.3-*g*/s acceleration rate were adopted as design criteria for the ignition system and motor thrust decay. When the actual spacecraft configuration is known better, these design values should be reassessed.

Another significant deviation from the nominal performance values of Figure 3 stems from the ignition delay in initiating burning of the 71.1-cm (28-in.)-diameter main motor by the *g*-dot igniter (typically 0.2 to 0.3 s in small-motor *g*-dot firings). Figure 4 shows the igniter pressure–time curve, and the results of the calculations for main motor pressure–time curves when ignition delays were assumed to be 0, 0.3, and 0.6 s. Results were obtained from an overall mass flow balance for the motor.

The ratio of the mass flow rate of the igniter to that of the main motor, a critical parameter in sustaining burning below the L^* combustion limit, is given in Figure 4 to show that its value lies above the

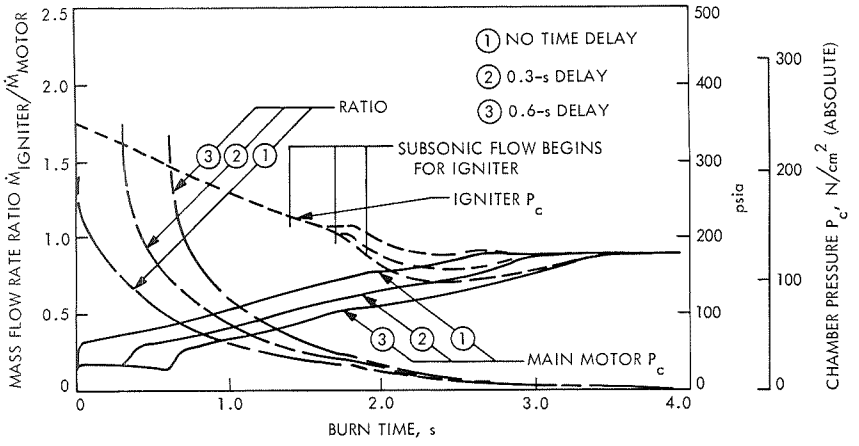

Figure 4. Mass flow rate ratio and chamber pressure vs burn time for main motor time delays of 0, 0.3, and 0.6 s

desired value of approximately one at the start of main motor burning for the given motor and igniter design conditions. Accelerations and acceleration rates fall within the 0.3 allowable values. Thus, it is believed that normal ignition delays will cause no difficulties in the firing of the demonstration motor D-2.

Internal Versus External Igniter System

There are advantages and disadvantages to an internal (torus) igniter compared to a 2- to 3-motor external igniter system, the more conventional arrangement. Figure 5 shows the two design concepts. The internal igniter was finally adopted, based on the following judgments.

The internal torus has the disadvantage that the main motor nozzle could become partially or completely plugged (disastrously) if uncontrolled breakup of the empty igniter case occurred during main motor burning. The problem appears to have been solved (1) by blocking the igniter into a cradle and cementing it to the nozzle phenolic insulation, and (2) through the use of a thermoplastic igniter case that will melt into a viscous glob around the nozzle and tend to wash away gradually into the stream of hot gases.

A second disadvantage results from the interface with the safe and arm mechanism, which, according to launch operations specifications, must be located outside the main motor. The safe and arm mechanism is relatively simple for external igniters where it could be located at the head end of the units with the initiator squibs. The use of Pyrofuze is being considered for the torus internal igniter in an effort to reduce the problem—or perhaps eliminate it.

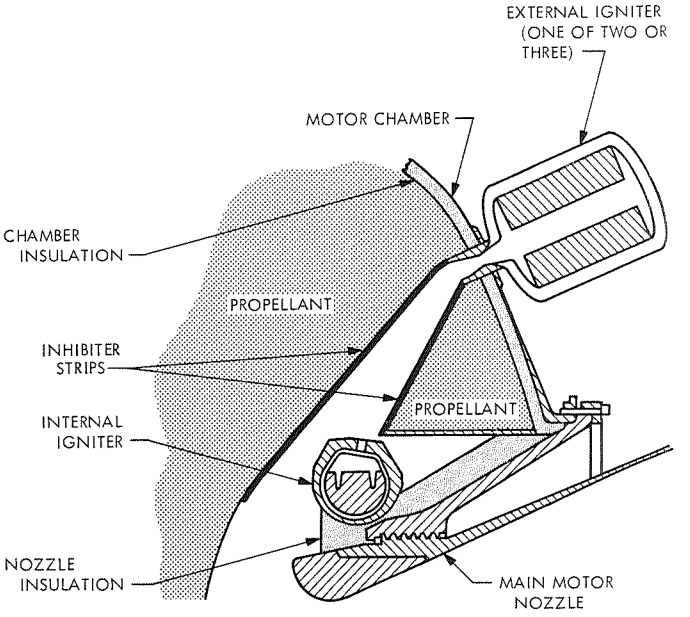

Figure 5. Internal (torus) igniter vs external igniter

Advantages of the internal igniter include (1) a required igniter hardware life of only 3 to 5 s compared to 140 s for the external igniters; (2) lower weight, fewer igniters and safe and arm mechanisms than the external system [the thickness of the insulation would be 1.02 cm (0.40 in.) in the external motor compared with 0.076 cm (0.03 in.) in the torus]; (3) flame spreading with 12 nozzles over the 71.1-cm (28-in.)-diameter charge surface should be better; and (4) the main motor case design and fabrication would be easier—no bosses on heat-treated chambers, no ports through the insulation and propellant, and fewer igniters to load with propellant.

Design of the *g*-Dot Igniter

Design features of the *g*-dot igniter are shown in Figure 6. Its outer diameter will be 28.7 cm (11.3 in.) and the small tube diameter will be about 5.33 cm (2.1 in.), with a wall thickness ranging from 0.3175 to 0.635 cm (⅛ to ¼ in.). The igniter case will be made of polycarbonate, a thermoplastic material, and will have 12 graphite nozzles equally spaced around the case and firing radially outward to promote flame spreading over the main propellant charge. Each nozzle will have a throat diameter of only 0.584 cm (0.23 in.); a propellant with only 2% aluminum has been selected in order to minimize changes in throat area and pressure from oxide deposition. The propellant weight is estimated at 1.07 kg (2.35 lb); the inert weight is the same. A layer of 0.076-cm (0.030-in.) rubber insulation on the inside, and perhaps on the outside, helps protect the plastic case for its 2½-s burning time. It has been designed with a safety factor of 2, based on limit loads.

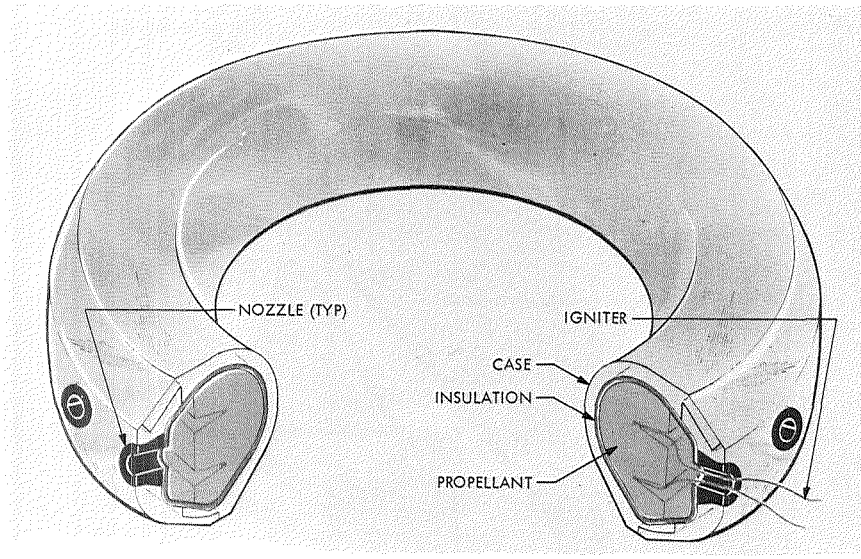

Figure 6. The g-dot igniter

Initiating Igniter Burning

The question of how best to initiate burning in the *g*-dot igniter in a hard vacuum is unanswered at this time. Vacuum ignition systems are available but an operationally simple system for a torus igniter poses something of a challenge.

Ignition, itself, involves the addition of external heat to the propellant at a rate such that its own subsequent decomposition and combustion produce sufficient heat to sustain further decomposition and burning when the external heat source is withdrawn. The process is complicated, especially in a vacuum, by (1) the propellant deflagration limit P_{dl}, that absolute pressure below which the propellant will not burn [about 2.07 to 2.76 N/cm^2 (3 to 4 psi) for the propellant under consideration]; (2) short residence times for the igniter combustion gases, i.e., low L^* motor values, so that there is insufficient time for heat transfer to the propellant to develop the required thermal gradient beneath the propellant surface, a necessary condition for sustaining burning; and (3) rapid decreases in pressure, i.e., high dp/dt, that can quench the burning—the rapid decrease may stem from the blowout of partial obstructions to gas flow or nozzle closures deliberately introduced earlier to raise the local pressure at the ignition surface above the P_{dl} value at which burning can be sustained.

The use of Pyrofuze wire is being considered for initiating burning in the igniter while in a hard vacuum. The Lockheed Propulsion Company and the Air Force Rocket Propulsion Laboratory have used Pyrofuze

successfully for igniting motors (Reference 2) but their experience appears to have been limited to firings at atmospheric pressure.

Pyrofuze itself is a coaxial wire with an aluminum core surrounded by a tube of palladium in intimate contact with the core. It has the unique property, when heated electrically or chemically to the 933 K (1220°F) melting point of aluminum, of instantly generating a large exotherm, under gas pressure or in a vacuum, due to the resultant alloying. Temperature in the wire alone can reach about 3030 K (5000°F), well above the 477 to 700 K (400 to 800°F) ignition temperature range of most solid propellants.

Figure 7 shows the cross section of the torus tube and location of the Pyrofuze, which is used here as a special type of hot wire igniter. A continuous braid of 8-strand 0.076- to 0.25-mm (3- to 10-mil) wire, or 0.025- to 0.127-mm (0.001- to 0.005-in.)-thick by 3.17-mm (0.125-in.)-wide Pyrofuze foil, would be buried just beneath the surface of the propellant slot. When current is passed through the wire, burning would be initiated throughout the charge circumference in the bottom of the slot so that hot combustion gases must pass over the walls of the propellant slot to reach the nozzle, thus promoting flame spreading and tending to raise the pressure locally near the line of ignition.

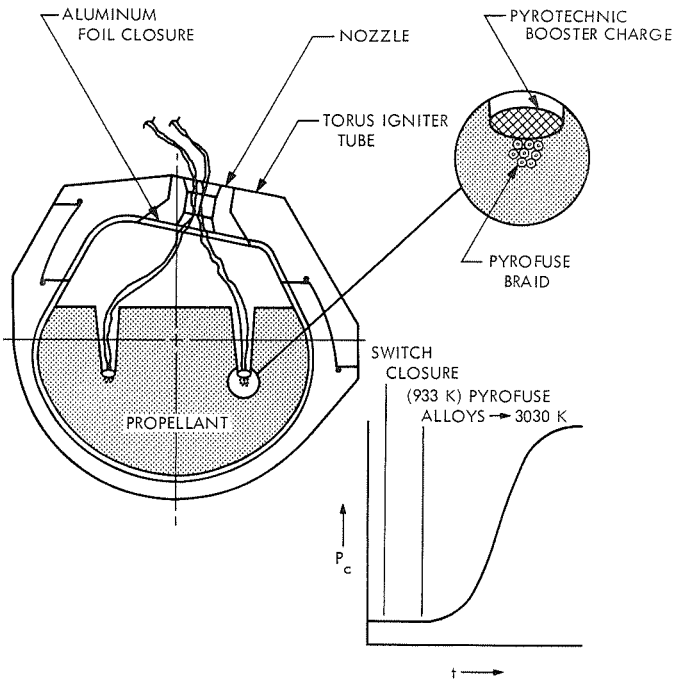

Figure 7. Initiation of g-dot igniter with Pyrofuze

Ten of the 12 nozzles in the torus will have thin aluminum foil as nozzle closures to raise the igniter L^* to 10.5 m (414 in.) for the two open nozzles and to increase the residence time of the hot gases in the igniter. To avoid abrupt drops in igniter pressure (high dp/dt) as the foil is ruptured, one nozzle will be closed with one layer of foil, three nozzles with two to three layers of foil, and six nozzles with three to five layers.

If the high igniter L^* and the nozzle closures fail to provide sustained burning from a hard vacuum start, the propellant will be (1) preheated locally by the Pyrofuze, below the propellant decomposition temperature of 450 to 477 K (350 to 400°F), by means of a low current for 5 to 30 s, then (2) ignited by triggering the Pyrofuze exotherm with a substantially higher current.

Finally, if all of the preceding approaches were to prove inadequate, a propellant that is more easily ignited than the polyether polyurethane JPL 540 Mod C will be substituted or, alternately, the use of a pyrotechnic booster charge or paste (Figure 7) will be used in the slot to provide more heat in a longer duration operation.

Processing and Igniter Testing

Although formation of the propellant charge in the torus case would appear to be difficult, practice tests have already revealed acceptable techniques. With a split mandrel of Teflon machined to form the finished propellant contour taking the place of the outer ring of the torus (Figure 7), propellant is vacuum-cast radially through the mandrel until a surplus fills the sprue. It is then cured for four days at 333 K (140°F), cooled, and the split mandrel removed. Before firing, the two sections of the torus are bonded together with an epoxy adhesive.

The future testing program should include: a proof–pressure test of the case, a qualification static firing of the igniter alone in the open, and one or more firings in an ignition test motor that duplicate the ignition phase of the D-2 motor. If practical, a simulated altitude firing of the ignition test motor should precede the firing of the demonstration motor D-2.

References

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2. Laufman, P. N., and Dilts, H. S., *Exothermic Bimetallic Ignition System,* AIAA Paper 69-425. AIAA 5th Propulsion Joint Specialist Conference, U. S. Air Force Academy, Colo., June 9–13, 1969.